



Infrared On-orbit RCC Inspection with the EVA IR Camera: Development of Flight Hardware from a COTS System

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ABSTRACT

In November 2004, NASA's Space Shuttle Program approved the development of the Extravehicular (EVA) Infrared (IR) Camera to test the application of infrared thermography to on-orbit reinforced carbon-carbon (RCC) damage detection. A multi-center team composed of members from NASA's Johnson Space Center (JSC), Langley Research Center (LaRC), and Goddard Space Flight Center (GSFC) was formed to develop the camera system and plan a flight test. The initial development schedule called for the delivery of the system in time to support STS-115 in late 2005. At the request of Shuttle Program managers and the flight crews, the team accelerated its schedule and delivered a certified EVA IR Camera system in time to support STS-114 in July 2005 as a contingency. The development of the camera system, led by LaRC, was based on the Commercial-Off-the-Shelf (COTS) FLIR S65 handheld infrared camera. An assessment of the S65 system in regards to space-flight operation was critical to the project. This paper discusses the space-flight assessment and describes the significant modifications required for EVA use by the astronaut crew. The on-orbit inspection technique will be demonstrated during the third EVA of STS-121 in September 2005 by imaging damaged RCC samples mounted in a box in the Shuttle's cargo bay.

INTRODUCTION

The Johnson Space Center has developed a damage-detection system on board the Orbiter to minimize risk and to ensure a safe return during re-entry [1,2]. Developed for STS-114 and STS-121, the first and second flights since the Columbia accident in 2003, the Space Shuttle's Remote Manipulator System (SRMS), the Orbiter Boom Sensor System (OBSS), and the Integrated Sensor Inspection System (ISIS) can be used to conduct on-orbit inspections, perform damage assessments, and identify mission critical damage to the Space Shuttle's Thermal Protection System (TPS) while minimizing impact to the mission schedule. The ISIS is a subsystem of the OBSS and provides multiple sensor systems that will inspect portions of the Orbiter's TPS after arrival on-orbit to determine if any damage has occurred.

The application of infrared thermography for damage detection was considered in the fall of 2003 and a concept study for adding an infrared camera system to the OBSS was tasked to NASA LaRC [3] and NASA GSFC. That study showed the feasibility of detecting cracks, holes, and silicon-carbide (SiC) loss by taking advantage of the natural thermal gradients induced in the RCC by solar flux and thermal emission from the Earth. However, the OBSS development was too far along to allow a feasible augmentation of an infrared system. Moreover, the OBSS may not be flown on missions after STS-121 because of volume and mass constraints.

In the summer of 2004, the Shuttle Program requested NASA LaRC lead a multi-center effort to consider the feasibility of detecting sub-surface damage to RCC using a handheld infrared camera. The use of flash



infrared thermography to detect both surface and sub-surface damage is a well-established technique. The RCC panels of the Orbiter are now inspected using flash thermography at NASA's Kennedy Space Center (KSC) pre and post-flight. The main question of the study was the ability to detect damage under passive, solar conditions without the use of an active heating source. As presented in this paper, the study showed that sub-surface damage to RCC could be detected using a FLIR S60 handheld camera under typical on-orbit solar conditions.

To demonstrate the inspection concept, a Demonstrate Technology Objective (DTO) was created for STS-121 that calls for imaging damaged RCC specimens during an EVA (spacewalk), and comparing results against ground data. This paper discusses the transformation of the FLIR S65 camera to an EVA-compatible, flight-certified system and presents the ground imagery that demonstrated the feasibility of a solar-based thermography inspection technique.

ON-ORBIT REQUIREMENTS

As shown in Figure 1, a crew member will be attached to the end of the Orbiter (or International Space Station (ISS)) arm and image the wing-leading edge of the Orbiter using the camera system from a nominal stand-off distance of 1.5m. Nominal operation procedures called for an astronaut to translate out to the site of interest, record a series of short movies of the target area, transfer the acquired data to a removable storage device, and then translate back inside the Orbiter. Data would be transferred to an Orbiter Payload and General Support Computer (PGSC) for subsequent downlink to the ground where the data would be processed and analyzed. Analysis and testing of damaged RCC material showed that the ability to detect sub-surface damage was improved when imagery was acquired during heating and cooling conditions. If it is not possible for the astronaut to shadow the target area during image acquisition, then the inspection time must be coordinated with sunset, or rely on shadowing from components of the ISS or the Orbiter itself.

The camera system was required to be hand-held and portable. Weight and dimension requirements were set at less than or equal to 2.0kg, and less than or equal to 254mm x 127mm x 127mm (LxWxH), not including display. The system was required to be battery powered and rely on radiative thermal transfer for heat dissipation. The volume requirement limited the size of effective radiator surfaces, and so power consumption was limited to 12W or less.

The boom-based infrared camera study showed that the spectral range of 7.5-13 μ m was acceptable for damage detection. Moreover, this range allows the use of uncooled microbolometer detector technology which results in a low-power, low-volume package.

Because developmental time constraints did not allow the development of a wireless data transfer system, on-board storage of radiometric data was required. Movies of up to 600 frames at frame rates up to 60 Hz were required to be recorded. Up to 6000 frames of recorded data must be stored on a removable storage device, such as a Compact flash card, to facilitate transfer to the ground through the Orbiter's downlink system.

The ability to acquire radiometric image movies were required to support necessary post-processing of the data required for damage detection. Expected scene temperature ranges were -40°C to 110°C. The boom-based study indicated that a noise equivalent delta temperature of at most 60mK at 30°C and a radiometric accuracy of $\pm 2.0^\circ\text{C}$ were sufficient for damage detection. Expected standoff distances required the instantaneous field of view (IFOV) to be 1.3mrad or less. Because the standoff distance will vary, the camera system must allow for an adjustable, motorized focus over the range of 0.3m to infinity.

The camera system will be mounted to the crew's Modular Mini Work Station (MMWS). To allow easier mobility by the crew during operation, the controls and display of the camera system were required to be separable from the camera unit by at least 0.5m. This separation allows the camera unit to be mounted separately on the MMWS from the control and display unit, and allows a crew member to point the camera in a separate direction from his nominal line of sight.

Over 275 requirements were levied on the camera system. Requirements covered environmental conditions: thermal, pressure, and radiation, as well as safety-related issues such as snag hazards, kickloads, entanglement hazards, and touch temperatures.

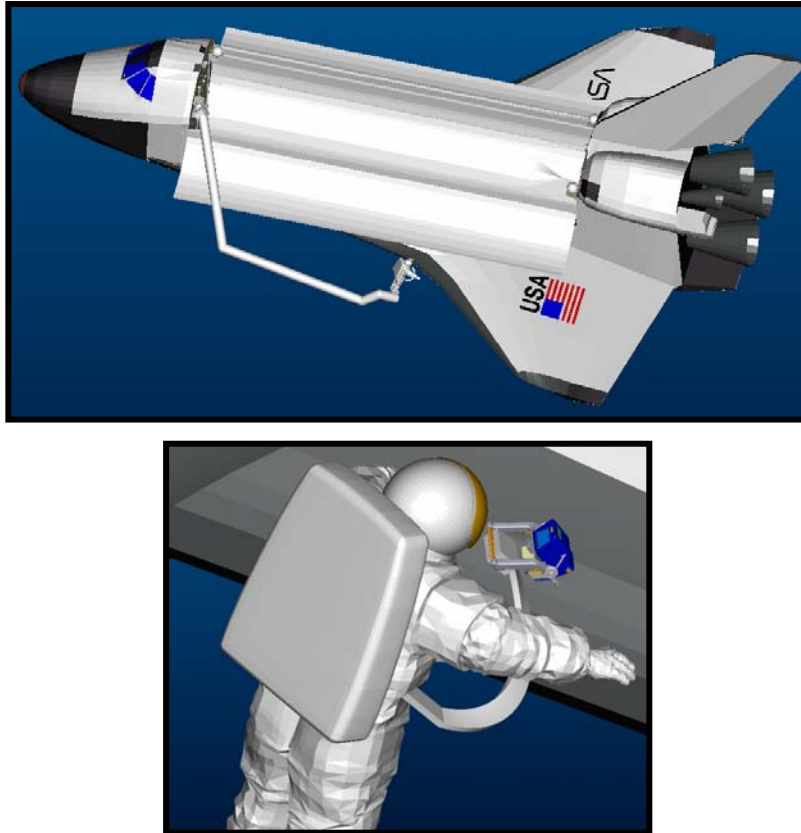


Figure 1. Imaging the wing-leading edge of the Orbiter during an EVA

COTS ASSESSMENT

The space environment imposes unique requirements on electro-optical hardware. In particular, the radiation, temperature, and pressure environment strain the operation and reliability of commercial products. In addition, the EVA environment imposes stringent requirements on user-interfaces, structural integrity, and overall safety. In the fall of 2004, a commercial FLIR S60 unit was subjected to a battery of tests to assess its space-flight capability and EVA use compatibility. Figure 2 shows the COTS S60 system. Within a 30-day period, a crew evaluation, thermal test, thermal-vacuum test, and radiation test were conducted on a commercial S60 unit. Functionality not required in the flight application was removed from the S60 and included the visible CCD imager, battery charge display, camera base controls, and viewfinder eye piece. Particular attention was placed on the operation of the liquid crystal display (LCD) as previous experience indicated issues with LCD displays at temperature extremes.



Crew Evaluation

As shown in Figure 3, the S60 was mounted on the MMWS and evaluated by crew members in a 1-g suited configuration. The remote control and display unit is mounted in front of the astronaut on the T-bar away from the camera base. The evaluation found that the FLIR S60 was largely usable "as-is." An extended joystick was recommended to allow easier use with the bulky EVA gloves and a sun-shade for the display unit (shown in Figure 3). The evaluation also determined that the camera base should allow attachment to the swing arm via a standard MMWS bayonet, and allow attachment to the body restraint tether (BRT). The BRT is a flexible arm that allows easy movement and is shown in the bottom right corner of Figure 3. While the COTS S60 controls were found acceptable, subsequent evaluations led to the replacement of the joystick with a 3x3 matrix of EVA-glove compatible buttons. Furthermore, the firmware of the camera was modified to allow one-button operation of key actions such as recording and data transfer.



Figure 3. Crew Evaluation of the FLIR S60 by NASA Astronaut Michael Gernhardt

Tenney Oven and Thermal-Vacuum Tests

As shown in Figure 4, the S60 was placed in a Tenney oven to determine the temperature range of the internal electronics over the -15°C to 50°C operational range of the commercial unit. The temperature extremes on the camera housing ranged from -12°C to 50°C, while the temperature extremes on the LCD varied between -10°C and +53°C. The hottest internal component ran 30°C over the camera housing temperature.



Figure 4. Tenney Oven Evaluation of the FLIR S60

The temperature data from the thirteen thermocouples installed in the S60 during the Tenney oven evaluation was used to design the thermal-vacuum test. The objective of the thermal vacuum test was twofold. The first objective was to assess the performance of the camera and liquid crystal display (LCD) in a vacuum environment. The second objective was to evaluate the thermal performance of the camera in a vacuum. The S60 was designed to take advantage of natural convection to cool the internal electronic components. However, in a vacuum environment, the system must rely on radiation and conduction to cool the electronics. In order to ensure that the camera would not overheat during the test, a conductive path was created from each of the high-powered electronic chips to the camera housing. Figure 5 shows a representative copper plate that was used to create a conductive path necessary to cool the electronic components.

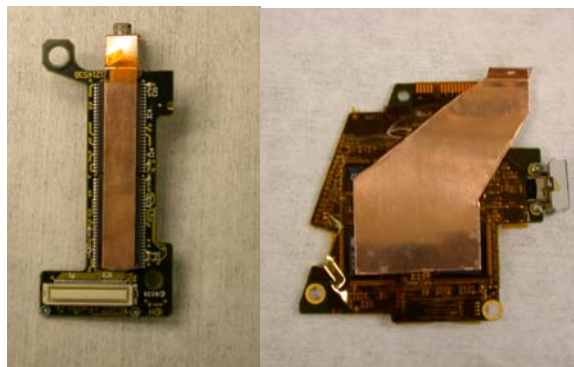


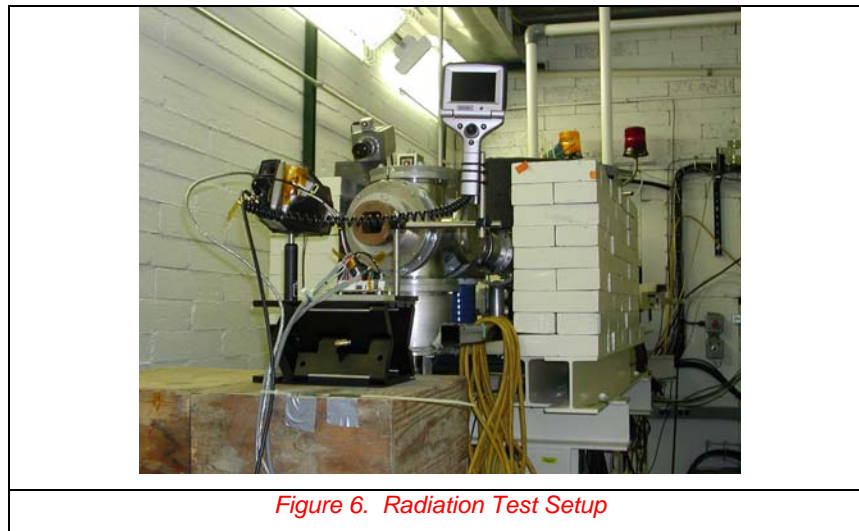
Figure 5. Example of Copper Thermal Straps

The addition of the conductive paths lowered the chip-to-case thermal resistance by approximately 63%. As was the case with the Tenney oven test, the camera was instrumented with 13 thermocouples in order to monitor the temperature throughout the test. The thermal vacuum shroud was set to maintain a temperature

of -75°C for the cold case and $+20^{\circ}\text{C}$ for the hot case at an ambient pressure of 0.04 Torr. The FLIR S60 operated successfully with the addition of the thermal "straps" and pads that were added to create a conductive path from the electronic components to the camera housing. The LCD operated at a temperature of -22°C with no noticeable degradation in the performance of the unit. In a -15°C ambient environment (manufacturer's minimum temperature limit) with the presence of air, the LCD reached a temperature of -10°C during the Tenney oven test. However, during the thermal vacuum test, the LCD operated approximately 12°C colder than the published limit. At the conclusion of the final test, the thermal vacuum portion of the assessment was terminated because the LCD high-voltage electronics appeared to begin arcing because of corona discharge. The chamber pressure was very near the corona area where the breakdown voltage of air is small. The high-voltage LCD backlight driver was replaced with a light emitting diode (LED)-based design in the flight unit.

Radiation Test

The fourth, and final, test performed in the COTS assessment portion of this project was a radiation test. This test was performed at the Indiana University Cyclotron Facility (IUCF) to assess the susceptibility to high-energy ionizing radiation for several elements of the S60 IR camera. It is imperative that hardware elements be able to operate in the environment for the duration of their missions. The two major elements of the ionizing radiation environment are the deposition of energy from Total Ionizing Dose (TID) and Single Event Effects (SEE) produced by high energy particles like protons and atomically heavier ions. The goal of the radiation testing was to establish estimates of the Mean Time Between Failures (MTBF) for each type of SEE detected for a given test article or electronic component. All of the testing was completed with a proton beam energy of 200 Mega-electron Volts (MeV). The test setup can be seen in Figure 6.



The normal beam diameter of approximately 6 cm was passed through various copper vignettes to adjust the size of the final beam allowed to radiate the test article. Nineteen beam positions were tested. Each position of the hardware tested (except positions 1, 7, 14, 15, 18, and 19) received a minimum TID of 590 Rads(Si) with no degradation in performance or single event effects. The other beam positions resulted in either single-event events or functional interrupts. Power latch-up events were observed and monitored by test support equipment. The accumulated on-orbit MTBF for the functional interrupt errors was 29.7 days which is sufficient for EVA use. The observation of latch-up events indicated that latch-up protection circuitry would be required to be installed in the flight camera system.

The assessment of the COTS S60 showed that the camera system was a good candidate for an EVA space-flight application. The performance of the camera, augmented with copper thermal straps, in a vacuum and at temperature extremes indicated that operation in the space environment was feasible. Furthermore, the ability of the camera to recover from each SEE, via a power recycle, demonstrated the feasibility of operating

the COTS electronics in the radiation environment of an EVA. Radiation testing also showed that a latch-up protection circuit would be required at the power interface.

FLIGHT UNIT SYSTEM DEVELOPMENT OVERVIEW

Figure 7 shows the engineering unit, built from a COTS S65 system, and the flight unit, built from a FLIR modified S65 unit. Major modifications included removal of un-needed functionality: Bluetooth interface, Firewire interface, high-temperature iris, visible imager, battery display, viewfinder, camera base controls, and camera housing. The retained components included the electro-optical assembly (microbolometer, focus motor, and lens) and four electronic boards. The COTS laser pointer was replaced with a ruggedized version that was better equipped for space-flight operation.

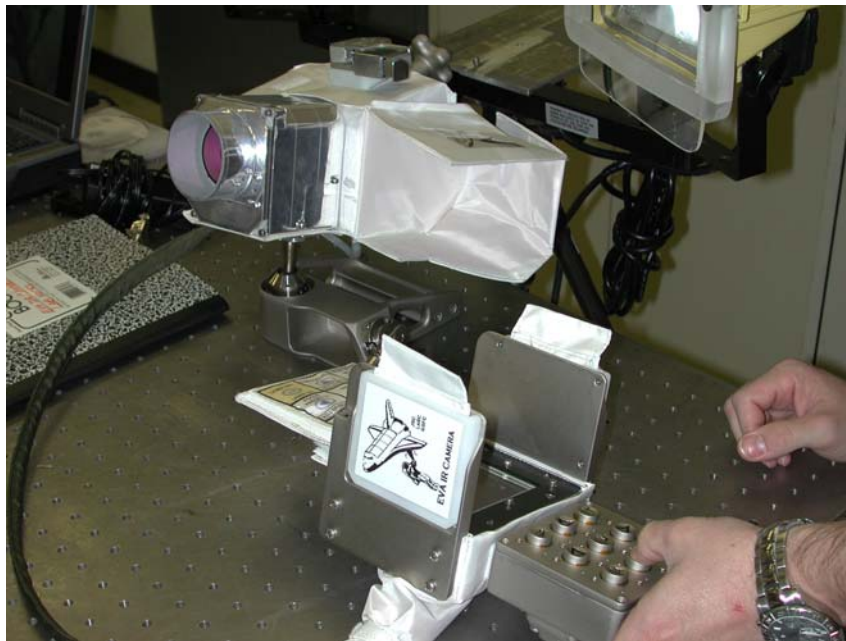


Figure 7a. Engineering Unit



Figure 7b. Flight Unit



The certification process for flying batteries in the manned space-program can be time intensive. Instead of certifying the COTS S65 battery, an interface card was designed to interface to the EVA Helmet Interchangeable Portable (EHIP) battery pack that is used in the helmet of the EVA suit. Already certified, the battery pack contains internal protection and has ample power to supply the camera for the required duration of an EVA (typically 6-8 hours). The interface card was designed to contain both the interface circuitry to the EHIP battery as well as provide latch-up protection.

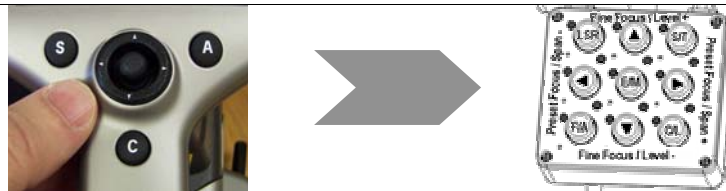
The cold cathode fluorescence lamp and associated high-voltage electronics was replaced with a LED-driven light pipe from Phlox Incorporated. This eliminated the corona issue and provided a brighter display. The new backlight driver electronics was combined with the COTS control interface electronics into a single board.

Further testing at JSC's Neutral Buoyancy Lab (NBL) with rapid-prototype mockups of the camera system refined the mechanical interfaces and mechanical package. The internal electronics and optics were enclosed in an aluminum housing that offered better structural protection against kickloads, and better thermal performance. The new housing mounted to a interface bracket through a ball joint. The ball joint allowed the crew member to orient the camera base in all three axes. The interface bracket contained both a interface to the BRT and a clockable bayonet that allows attachment to the MMWS and swing arm. The housing also provided the mechanical interface to the EHIP battery, and a slidelock on top of the camera base that allows attachment of the remote control unit (RCU) via a bayonet interface.

The entire suite of environmental series of tests was repeated on the engineering unit in the spring of 2005. The successful radiation test demonstrated the successful operation of the latch-up circuitry, and the successful thermal-vacuum test validated the thermal model and thermal design (copper straps and Thermagon sil-pads). In addition, survival heaters were added to the display and control unit and base unit to provide heat when the camera is not in the on position. A thermal blanket covers the unit and protects the lens when the system is not in use. During operation, part of the blanket folds back to expose the lens and expose key radiator surfaces as shown in Figure 7. Vibration and shock testing verified the design and workmanship of the mechanical housing, interfaces, and electronic assemblies.

Interface Modifications

As shown in Figure 7, the remote control of the unit consists of nine buttons that meet EVA glove interface requirements. Working with FLIR, a modified version of the S65 was manufactured for NASA that contained firmware modifications to allow one-touch activation of certain key events such as movie recording and data transfer. Figure 8 shows the button-function mapping. Simulations of the EVA operation with the crew showed that it was important to activate movie recording with a single button activation.



SSS Label	Joystick Left Right Joystick Up Joystick Down Joystick Center (SELECT) S A C Laser Toggle										
EVA IR Camera Label	◀ Focus Preset +	▶ Focus Present -	▲ Focus +	▼ Focus -	E/M Enter /Menu Mode	S/T Start or Stop /Transfer	F/A Flat Field Correction /Abort	C/L Cancel or Change mode /Level Span	LSR Laser Off /Laser On	Mode	Function
Button Action							Push & Hold (A&C)	Push & Hold (A&C)		ALL	Default (Master) Reset
							Push			Normal	Flat-field Correction (NUC) (Auto focus disabled)
	Push (decrement)	Push (increment)								Normal	3'-6'-9'-12'-∞
			Push	Push						Normal	Focus
					Push					Normal	Start Movie Record (up tp N frames)
					Push Once (after movie start)					Normal	Stop Movie Record
							Push & Hold			Normal	Dumps (Erases) Recorded Images
							Push & Hold			Normal	Transfer Movie to Flash Card
							Push & Hold (after movie transfer start)			Normal	Abort Movie Transfer
								Push & Hold		Normal	Change to Level/Span Mode
					Push & Hold					Normal	Enter Menu Mode
	Push	Push	Push	Push						Menu	Move Field, Toggle Parameters
					Push					Menu	Accept Parameter Changes
							Push			Menu	Exit, or Cancel
	Push	Push								Level/Span	Change Span
			Push	Push						Level/Span	Change Level
								Push		Level/Span	Change to Normal Mode
									Push & Hold	Any	Laser On (Not a FLIR requirement)
									Push	Any	Laser Off (Not a FLIR Requirement)

Figure 8. Remote Control Button Mapping

Focusing on the uniform RCC panels on-orbit presented a challenge to the camera focus as the material contains a small amount of contrast. The motorized focus operation was modified to incorporate a series of pre-set focus distances that are controlled by the crew member with the left-right arrow buttons. The focus distance setting was also displayed to the user on the LCD. These changes allowed the crew member to judge the standoff distance from the target area, set the nominal focus distance to the judged distance, and then use the fine adjustment for small adjustments.

RCC INSPECTION UNDER SOLAR CONDITIONS

The feasibility of detecting sub-surface damage under on-orbit solar conditions was explored by imaging damaged RCC samples using quartz lamps to simulate the expected solar flux. The maximum expected absorbed solar flux on-orbit is approximately 1100W/m². A pair of quartz lamps were used in a laboratory environment (ambient temperature and pressure) to simulate expected solar flux. By measuring the temperature profile in RCC and knowing the material properties of RCC (thermal conductivity), the lamp power settings were calibrated to solar flux levels. In addition, the measured temperature profiles were compared to a model simulation of RCC and found to be in good agreement. Figure 9 shows the test setup.

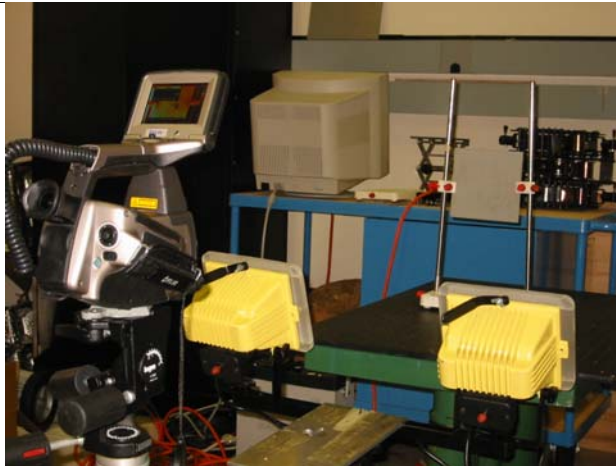


Figure 9. Imaging Damaged RCC Under Simulated Solar Conditions

Figure 10a shows three images of a damaged RCC sample. The sample was subjected to a foam impact at Southwest Research Institute. A visual image of the sample, leftmost picture, shows SiC loss in a downward diagonal line in the upper left corner. An ultrasound image of the same sample, middle picture, reveals a $\sim 9\text{in}^2$ delamination in the center of the sample. A processed IR image, rightmost picture, acquired by a COTS S60, shows a delamination area similar to the ultrasound image. A 30Hz rate movie was acquired by the S60 for 20 seconds. Data was recorded during the heating and cooling cycle (lamps turned off). The movie was then processed using the Principal Component Analysis (PCA) time-series technique. PCA has shown to be effective to improve detection capability of sub-surface defects. The PCA window was centered at frame number 150 where the quartz lamps were extinguished. Acquiring the image movie during the cooling cycle, or while the sample is shadowed, helps distinguish damage from non-uniform heating and minimizes effects from reflections. PCA takes advantage of the difference in cooling rates of the nominal RCC and damaged area.

Figure 10b shows the average signal levels per frame, and the processed IR image at a simulated flux level of 225W/m^2 . The red line is an average from pixels in the damaged area and the green line is a average from an area of corresponding size outside the damaged area. Though the signal to noise level is reduced, the damaged area is still detectable.

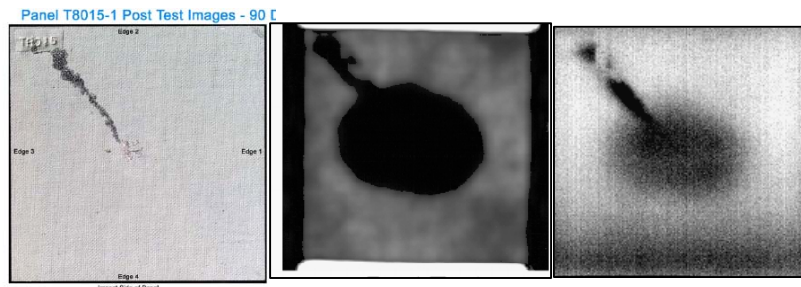


Figure 10a. Damaged RCC specimen: visible image, ultrasound image, processed IR image at 1100W/m^2

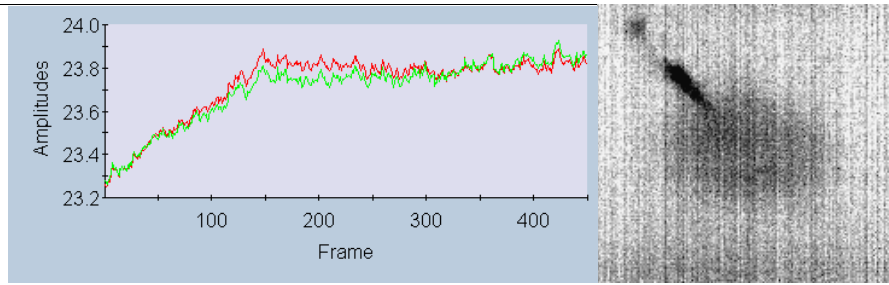


Figure 10b. Signal level per frame and processed IR image at 225 W/m^2

Figure 11 shows a prediction of the on-orbit temperature gradient through RCC panel 9. Certain periods during the 90 minute orbit contain higher through-gradients than others and indicate the presence of larger heating and cooling rates. These periods are desirable times to perform damage inspection.

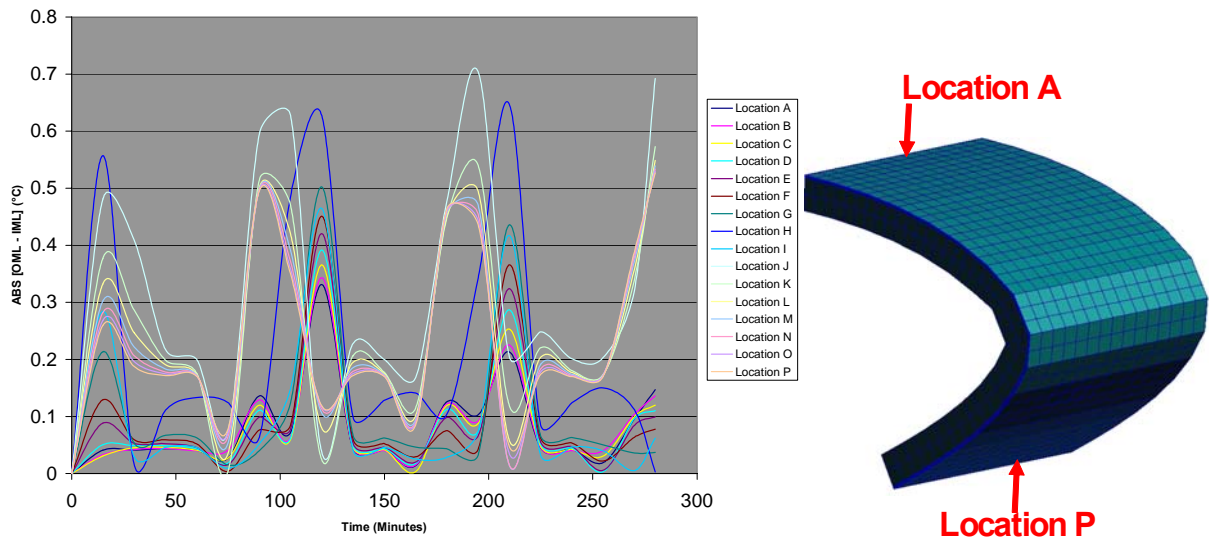


Figure 11. Expected On-orbit Through Temperature Gradient for RCC Panel 9



SUMMARY

In November 2004, NASA's Space Shuttle Program approved the development of the Extravehicular (EVA) Infrared (IR) Camera to test the application of infrared thermography to on-orbit reinforced carbon-carbon (RCC) damage detection. A multi-center team composed of members from NASA's Johnson Space Center (JSC), Langley Research Center (LaRC), and Goddard Space Flight Center (GSFC) was formed to develop the camera system and plan a flight test. At the request of Shuttle Program managers and the flight crews, the team accelerated its schedule and delivered a certified EVA IR Camera system in time to support STS-114 in July 2005 as a contingency. The development of the camera, led by LaRC, was based on the FLIR S65 handheld infrared camera. The feasibility of detecting sub-surface damage using IR thermography was established from experimental data using damaged RCC samples under predicted solar conditions. Through a series of environmental tests and crew evaluations, a design concept for a hand-held camera based on the FLIR S65 was developed. A certified flight unit was built and delivered for STS-114 in July 2005, eight months after project approval.

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